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RESEARCH MEMORANDUM

FLIGHT INVESTIGATION OF PENTABORANE FUEL IN ROCKET BOOSTED 9.75-INCH-DIAMETER RAMJET ENGINE WITH CONVERGENT-DIVERGENT EXHAUST NOZZLE

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RESEARCH MEMORANDUM

FLIGHT INVESTIGATION OF PENTABORANE FUEL IN ROCKET BOOSTED 9.75-INCH-DIAMETER RAMJET ENGINE WITH CONVERGENT-DIVERGENT EXHAUST NOZZLE

By John H. Disher

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SUMMARY

A flight test of a pentaborane-fueled air-launched ramjet engine with convergent-divergent exhaust nozzle has been made. The engine was boosted to a Mach number of about 2.00 by a small internally housed rocket; the ramjet then accelerated to a maximum free-stream Mach number of 3.02 at an altitude of 29,500 feet. During acceleration, to the engine design-point Mach number of 2.4, pentaborane illustrated the ability to resist combustor blowout during severe inlet-buzz conditions. A maximum thrust-minus-drag coefficient of 0.81 was reached at a free-stream Mach number of 2.4. The minimum specific fuel consumption of 1.54 pounds of fuel per hour per pound of thrust minus drag was also observed at Mach number of 2.4.

The liquid boron oxide exhaust products had no discernible adverse effect on engine thrust.

INTRODUCTION

After initial laboratory and test-stand investigations of pentaborane as a ramjet fuel (refs. 1 and 2), flight investigations of the fuel were undertaken at the NACA Lewis laboratory in 1953. Results of three flight tests of this fuel in a 9.75-inch-diameter ramjet engine at Mach numbers to 2.06 are reported in references 3, 4, and 5. These three flights were made on a configuration with a 1.8 free-stream Mach-number design point and with a straight-pipe combustor. These models were launched from a carrier airplane at altitudes of about 35,000 feet and at a subsonic Mach number. During descent the engines then accelerated to supersonic velocity with the aid of gravity. These three flights successfully demonstrated that pentaborane could be handled safely under actual flight conditions and confirmed theoretical and test-stand performance figures for an engine with a sonic discharge nozzle.

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Concurrent with the flight investigation of these engines, the theoretical performance of pentaborane in a convergent-divergent nozzle was being considered. Calculations indicated that with a convergent-divergent nozzle an appreciable exit momentum loss might occur under some conditions because of the liquid exhaust products of pentaborane combustion. These losses would occur if the liquid exhaust products were not in velocity and thermal equilibrium with the exhaust gases. In order to investigate this effect in flight and to evaluate the over-all performance of a pentaborane-fueled ramjet at Mach numbers between 2 and 3, the present flight investigation was undertaken. An engine with a double-oblique-shock-type inlet of a 2.4 free-stream Mach-number nominal design point was chosen for the flight evaluation. Because a convergent-divergent nozzle precluded engine self-acceleration through the transonic Mach-number range, even with the aid of gravity, the engine was rocket boosted to an acceptable ramjet take-over Mach number. 4556

The results of a single flight with this engine configuration are presented herein.

APPARATUS AND PROCEDURE

Ramjet Configuration

A photograph of the launching plane used for this investigation is shown in figure 1, and a sketch of the ramjet configuration is shown in figure 2. The engine has a combustion chamber with a 9.75-inch diameter and an exit nozzle with an 8.25-inch throat expanding back to a 9.75-inch exit diameter. The exit nozzle is detailed in figure 3. The expansion half angle is 12°, and the contour radius of the nozzle throat is $4\frac{1}{8}$ inches. The combustion chamber and nozzle were fabricated from 1/16-inch Inconel. The nozzle was enclosed in a 9.75-inch-diameter cylindrical shroud of 1/32-inch Inconel to minimize drag.

A small rocket (Thiokol T-55) was mounted in the combustion chamber for boosting the engine from the launching velocity to the ramjet take-over Mach number. The rocket was mounted in a small aluminum casing (fig. 4), which rolled on aluminum tracks in the combustion chamber. The booster was held in place by a shear pin designed to break at an 800-pound load. The rocket was ignited by a 5-second-delay squib, which was energized as the engine was released from the carrier plane. Upon rocket ignition, the shear pin broke, and the rocket was then held in by its own thrust. Upon rocket burnout, the internal drag forced the unit out of the chamber. As the rocket ejected, it released the ramjet fuel by pulling a lanyard. Although pentaborane would probably ignite spontaneously, a small ignition source was provided. The igniter consisted of two small chemical flares ignited by 7-second-delay squibs. The 7-second delay corresponded to the approximate fuel release time.

The flameholder and fuel injector (fig. 5) were similar to those used in the final unboosted-flight engine (ref. 3). The fuel system was similar to that used in reference 3 as was the eight-channel telemeter. The inlet of the engine was of the double-oblique-shock type with cone half angles of 22° and 35° and was similar to that used in references 6, 7, and 8. The inlet lip was positioned to intercept the oblique shocks at a free-stream Mach number of 2.4. The inlet provided no internal contraction. The inner body of the engine was formed of 0.05-inch aluminum. The forward $38\frac{3}{8}$ inches of the external shell was formed of 0.09-inch aluminum; the next $17\frac{1}{8}$ inches was formed of 0.064-inch aluminum; and the remainder of the external shell of the engine was formed of 0.062-inch Inconel.

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The weight of the engine with the booster was 206 pounds. The center of gravity with the booster installed was located 60.75 inches downstream of the diffuser lip. Engine weight without booster and with 8.78 pounds of fuel was 150 pounds.

CZ-1 back

Instrumentation

The instrumentation used for testing consisted of the following:

(1) Free-stream static pressure.

(2) Free-stream total pressure.

(3) Total minus static pressure at station 2 (20 in. downstream of lip).

(4) Free-stream total pressure minus diffuser-exit total pressure at station 3 (64 in. downstream of inlet).

(5) Nozzle static pressure 2.2 inches from nozzle exit, station 6a.

(6) Nozzle static pressure 1/4 inch from nozzle exit, station 7a.

(7) Fuel pressure minus diffuser-exit total pressure.

(8) Axial acceleration.

The total pressures at stations 2 and 3 were measured by slotted averaging rakes. Static pressures were measured with flush wall orifices.

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CALCULATION PROCEDURE

The engine performance was calculated as follows: Atmospheric conditions were obtained by a radiosonde survey immediately following the flight. The model Mach number was computed from the total- and static-pressure measurements on the pitot-antenna probe and checked by radar and acceleration integration. Engine thrust minus drag was calculated as the product of instantaneous engine weight and engine axial acceleration (exclusive of gravity component). Actual fuel flow was calculated from measured fuel pressure and a preflight fuel-spray-bar calibration. 4556
Ideal fuel flow was determined as the fuel rate required to produce the observed exit temperature at a given flight condition, with 100-percent combustion efficiency. Engine airflow was calculated from the pressure measurements at station 2 and from the free-stream total temperature. With supercritical flow at Mach numbers above about 2.5, engine airflow was computed from free-stream conditions and inlet capture area.

Engine-exit total pressure was calculated from measured diffuser-exit total pressure. In this calculation, the known flameholder total-pressure loss of 1.2 dynamic heads and the theoretical combustion momentum pressure losses are subtracted from the measured diffuser-exit total pressure in order to obtain combustor-exit total pressure. The combustor exit total pressure was assumed constant through the exhaust nozzle.

The combustor exit total temperature was also assumed constant through the exhaust nozzle and was calculated at the nozzle throat by use of the continuity equation. For this calculation, the throat Mach number is assumed equal to unity, and all other quantities in the equation except temperature are known. Inasmuch as the gas properties are a function of temperature, an iteration process is required.

Ordinarily in a flight test of this type, the internal thrust of the engine is calculated by assuming an exit-nozzle coefficient based on cold-air tests of similar geometry nozzles. With the thrust so calculated, the external drag of the engine may then be calculated by subtracting the quantity (thrust minus drag) obtained by acceleration measurements from the thrust. In this case, however, it was desired to evaluate the exit-nozzle coefficient. Therefore, the external drag was computed from reference data and was added to the measured quantity (thrust minus drag) in order to obtain thrust. The exit-nozzle coefficient was then evaluated by comparing the thrust plus engine-inlet momentum with theoretical jet thrust for the existing nozzle pressure ratio and area ratio (ref. 9).

Thrust, thrust-minus-drag, and drag coefficients are based on a reference area of 0.52 square foot.

Thermodynamic data for the pentaborane and exhaust products were obtained from reference 10.

RESULTS AND DISCUSSION

Histories of the engine acceleration, Mach number, and altitude are presented in figures 6. The engine was launched at an altitude of 42,200 feet and a Mach number of 0.70 (fig. 6(c)). At 5.3 seconds after release the booster rocket ignited and accelerated the engine to a Mach number of 1.98 in 7.6 seconds (fig. 6(b)). The rocket then separated, and the ramjet ignited at an altitude of about 41,000 feet. Shortly after ignition, the engine inlet went into buzz, and engine thrust diminished as evidenced by an acceleration dropoff at 8.4 seconds (fig. 6(a)). The engine continued to buzz until it reached the design-point Mach number of 2.4 at 14.8 seconds. The altitude at this time was 35,000 feet. Continued engine operation under buzz conditions indicates one of the advantages of the boron hydride fuels, namely, excellent combustion stability. Previous experience with hydrocarbon fuels had shown that blowout would occur soon after a severe buzz condition was entered.

A part of the telemeter record showing engine acceleration variation during buzz is shown in figure 7. The buzz frequency varied from about 17 cycles per second at a Mach number of 2.0 to 10 cycles per second at a Mach number of 2.4. The amplitude of the thrust variation during buzz was about 1200 pounds at a Mach number of 2.4. With cessation of buzz, the engine axial acceleration increased to approximately 6 g's (exclusive of gravity component). The engine then accelerated to a peak Mach number of 3.02 at 20 seconds after release. The axial acceleration steadily decreased as the engine design point was exceeded, until at about 19 seconds, the acceleration rapidly diminished to zero, and the telemeter signal terminated shortly thereafter.

Free-stream static pressure and free-stream static and total temperatures are presented in figure 8.

Combustor-inlet Mach number, total-temperature ratio across the combustor, and equivalence ratio (fuel-air ratio divided by 0.0763) are presented against free-stream Mach number in figure 9. The curve showing data during engine buzz is dashed to indicate approximate values. The actual and ideal equivalence ratios are also shown. The ideal equivalence ratio is that required to produce the existing total-temperature ratio with 100-percent combustion efficiency. Actual equivalence ratios were not obtained during the latter part of the flight because of a malfunction in the fuel pressure measurement. The equivalence ratios and exhaust-gas temperature throughout the flight were such that the boron oxide exhaust products were in the liquid state.

The diffuser pressure recovery measured near the subsonic diffuser exit, station 3, is presented in figure 10. At a Mach 2.0, before the onset of inlet buzz, the pressure recovery is about 0.85, and at a Mach 2.4 the recovery is about 0.79. The theoretical recovery for these conditions is 0.95 and 0.89, respectively. Most of the decrement from theoretical values is attributed to friction in the subsonic diffuser. With the small engine size, it was necessary to make the subsonic-diffuser passage quite long in order to accomodate telemeter and fuel-system equipment in the centerbody. In references 6 and 7, 16-inch-diameter engines, utilizing the same type inlet, realized theoretical recovery values shortly downstream of the inlet.

As the engine accelerates beyond the design point, supercritical flow exists and the pressure recovery falls off rather rapidly.

The over-all engine total-pressure recovery is calculated to equal about 89 percent of the value measured at station 3. The measured nozzle static pressures divided by calculated exit total pressure are plotted against the nozzle area ratio in figure 11. Also shown is the theoretical pressure distribution for one-dimensional isentropic flow and the experimental pressure distribution for a typical nozzle having a thrust coefficient of about 0.98 (ref. 9). The agreement between the reference data and the measured wall pressures in the present test indicate that the liquid boric oxide exhaust products did not cause a discernible loss of momentum in the divergent section of the nozzle.

The engine thrust-minus-drag coefficient is presented as a function of Mach number in figure 12. At a Mach number of 2.0, before the onset of buzz, a value of 0.51 was observed. The values during buzz represent a mean of the cyclic thrust variation. These mean values were obtained by planimeter from the acceleration against time oscillograph record and were checked by differentiating free-stream velocity with respect to time. At a Mach number of about 2.4, the thrust-minus-drag coefficient increased from 0.55 to 0.81 with cessation of buzz. As the engine accelerated beyond design, the thrust-minus-drag coefficient progressively decreased to about 0.24 at a Mach number of 2.95. At this point a more rapid decline in thrust started, and thrust-minus-drag coefficient fell to about zero at a Mach number of 3.02, when telemeter transmission ceased.

In order to evaluate apparent exhaust-nozzle performance, the engine jet thrust was calculated and compared with the ideal jet thrust at near the design-point Mach number.

At a free-stream Mach number of 2.41 (15 sec), the engine mass-flow ratio was 0.89 (ratio of free-stream tube area of internal airflow to projected lip area). The measured thrust-minus-drag coefficient was 0.81.

The external-drag coefficient for the engine is estimated to be 0.15±20 percent for this flight condition. Thus, the engine thrust coefficient is 0.96±0.03. The free-stream momentum coefficient corresponding to a 0.89 mass-flow ratio is 1.22 and the jet thrust coefficient thus equals 2.15 to 2.21. The ideal jet thrust coefficient for these conditions is calculated to be 2.19. Thus, the ratio of actual to ideal jet thrust agrees with a cold-flow value of 0.98 (ref. 9) within the data scatter.

The engine specific fuel consumption, based on thrust minus drag, is shown in figure 13. At the design-point Mach number the specific fuel consumption was 1.54 pounds of fuel per hour per pound of thrust minus drag. The ideal value at the same conditions was approximately 1.45. The close agreement of the actual and ideal fuel consumption indicates little if any thrust loss due to the liquid oxide exhaust products.

At Mach numbers above design, the actual and ideal specific fuel consumption increased appreciably. At Mach numbers above 2.9, as the thrust minus drag approached zero, the specific fuel consumption based on thrust minus drag would of course approach infinity.

The calculated specific fuel consumption for a conventional hydrocarbon fuel operating under similar conditions would be 40 percent greater than the values observed for pentaborane during this test.

SUMMARY OF RESULTS

The flight test of a pentaborane-fueled, air-launched, 9.75-inch-diameter ramjet engine with convergent-divergent exhaust nozzle has provided the following results:

1. After boost by a small internally housed rocket to a Mach number of about 2.00, the ramjet accelerated to a maximum Mach number of 3.02 at an altitude of 29,500 feet.

2. The engine remained ignited and accelerated continuously while operating under severe inlet-buzz conditions between Mach numbers of 2.00 and 2.40.

3. A maximum thrust-minus-drag coefficient of 0.81 was observed during the flight. This value occurred near the design-point Mach number of 2.4 at an altitude of 35,000 feet. Maximum engine acceleration of approximately 6 g's also occurred at this condition.

4. The diffuser total-pressure recovery including subsonic diffuser losses was 0.85 at a Mach number of 2.0, and 0.79 at a Mach number of 2.4.

5. No discernible adverse effects of liquid boric oxide exhaust products on engine thrust were observed.

6. The engine specific fuel consumption reached a minimum value of 1.54 pounds of fuel per hour per pound of thrust minus drag. The calculated specific fuel consumption for a conventional hydrocarbon fuel at similar thrust conditions was 40 percent greater than this minimum value.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 3, 1957

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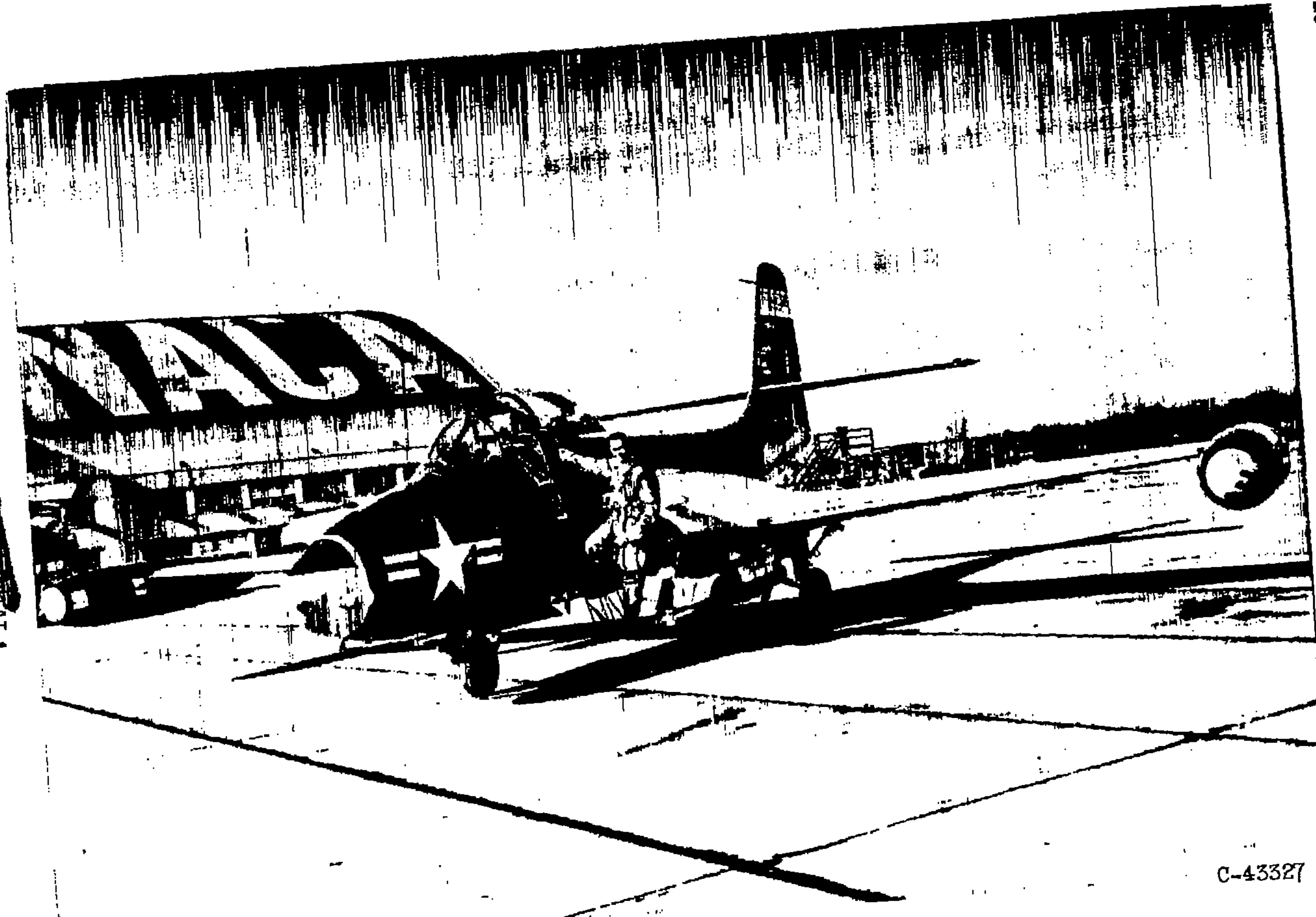

Figure 1. - F2H-2B airplane used for launching model.

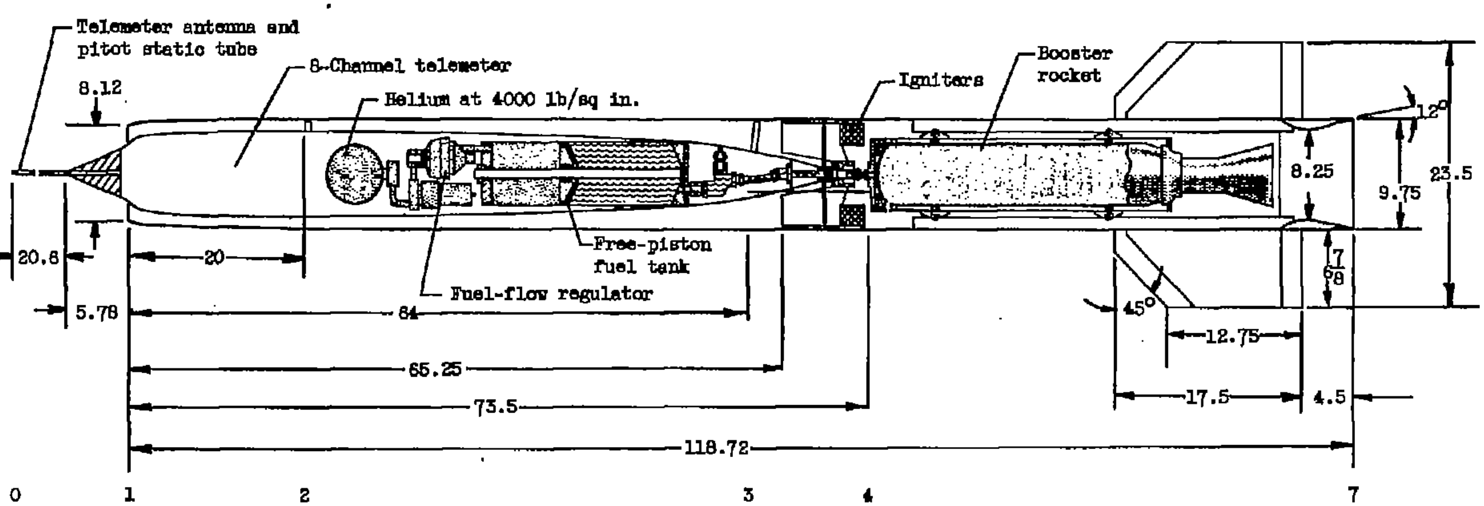

Figure 2. - Sketch of rocket boosted $9\frac{3}{4}$-inch-diameter ramjet with convergent-divergent exhaust nozzle. (All dimensions in inches.)

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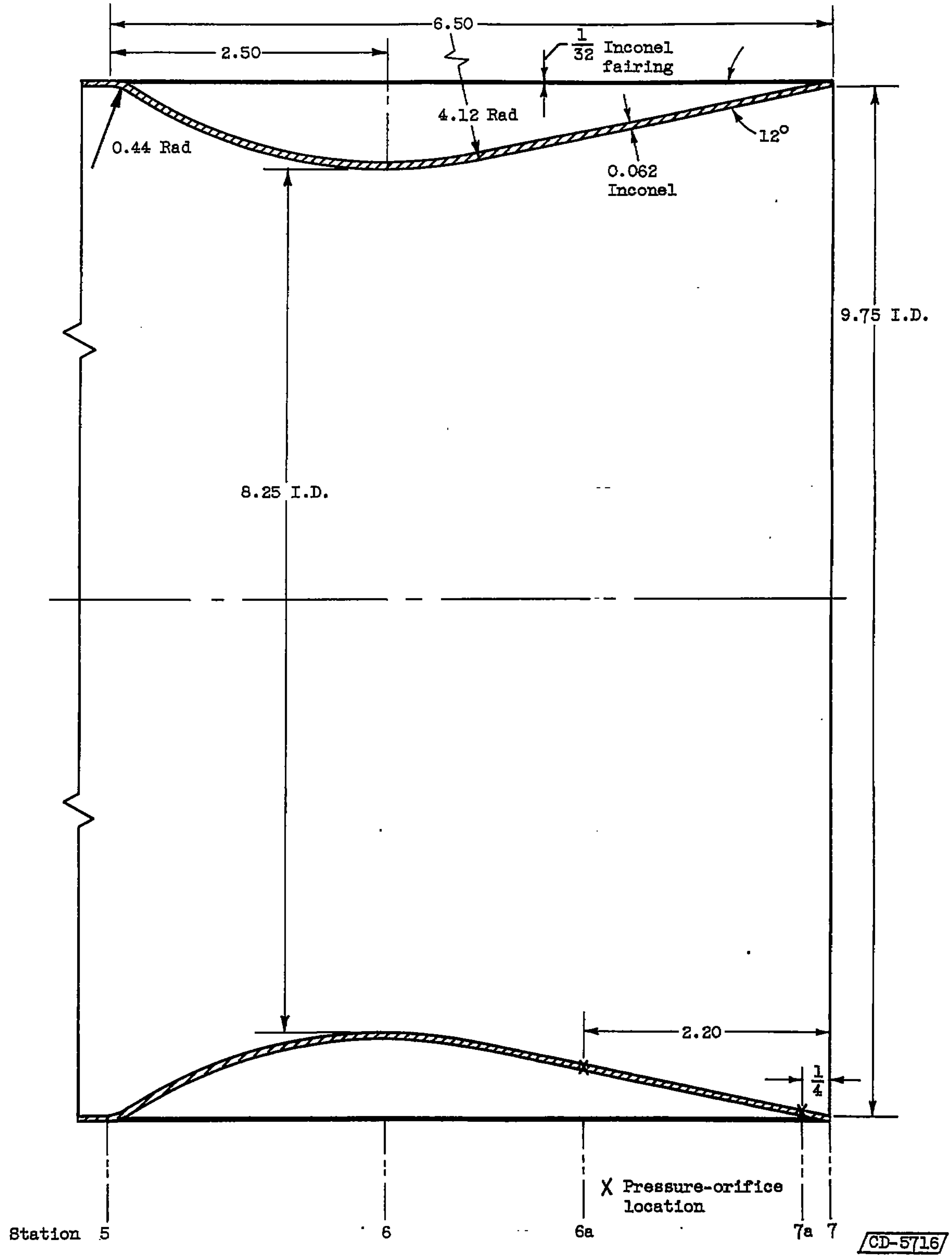

Figure 3. - Detail sketch of exit nozzle. (All dimensions in inches.)

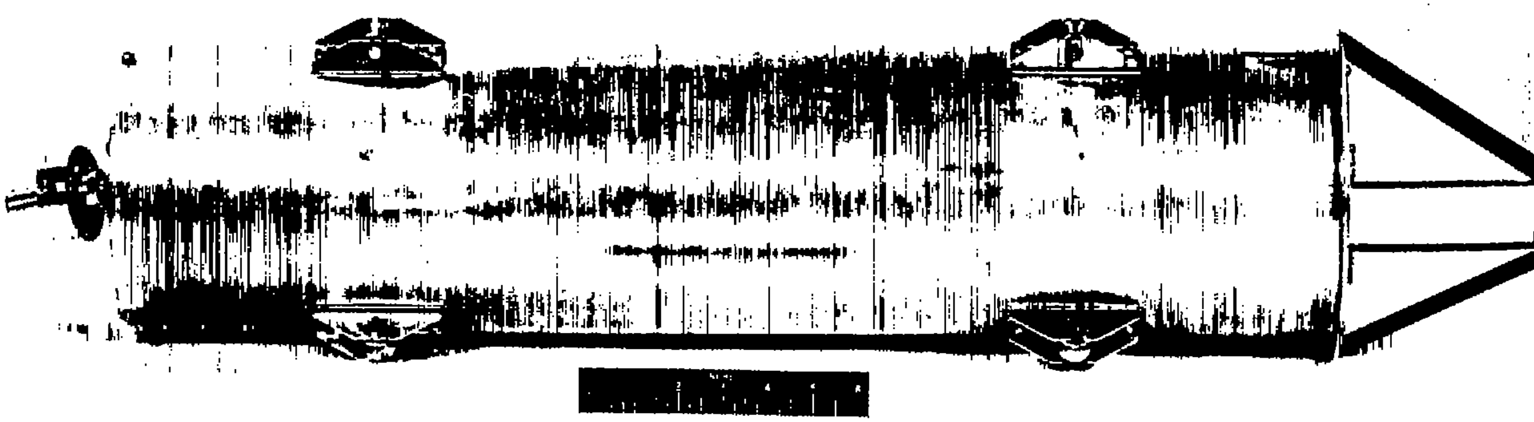

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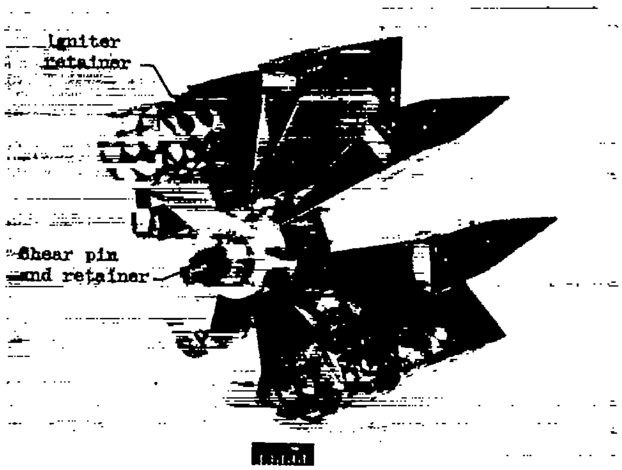
Figure 4. - Booster casing.



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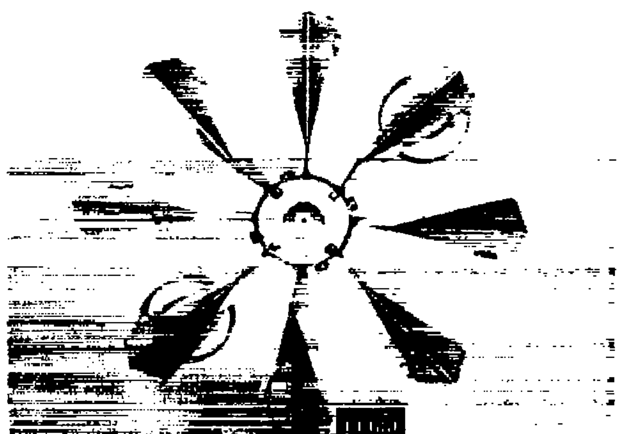

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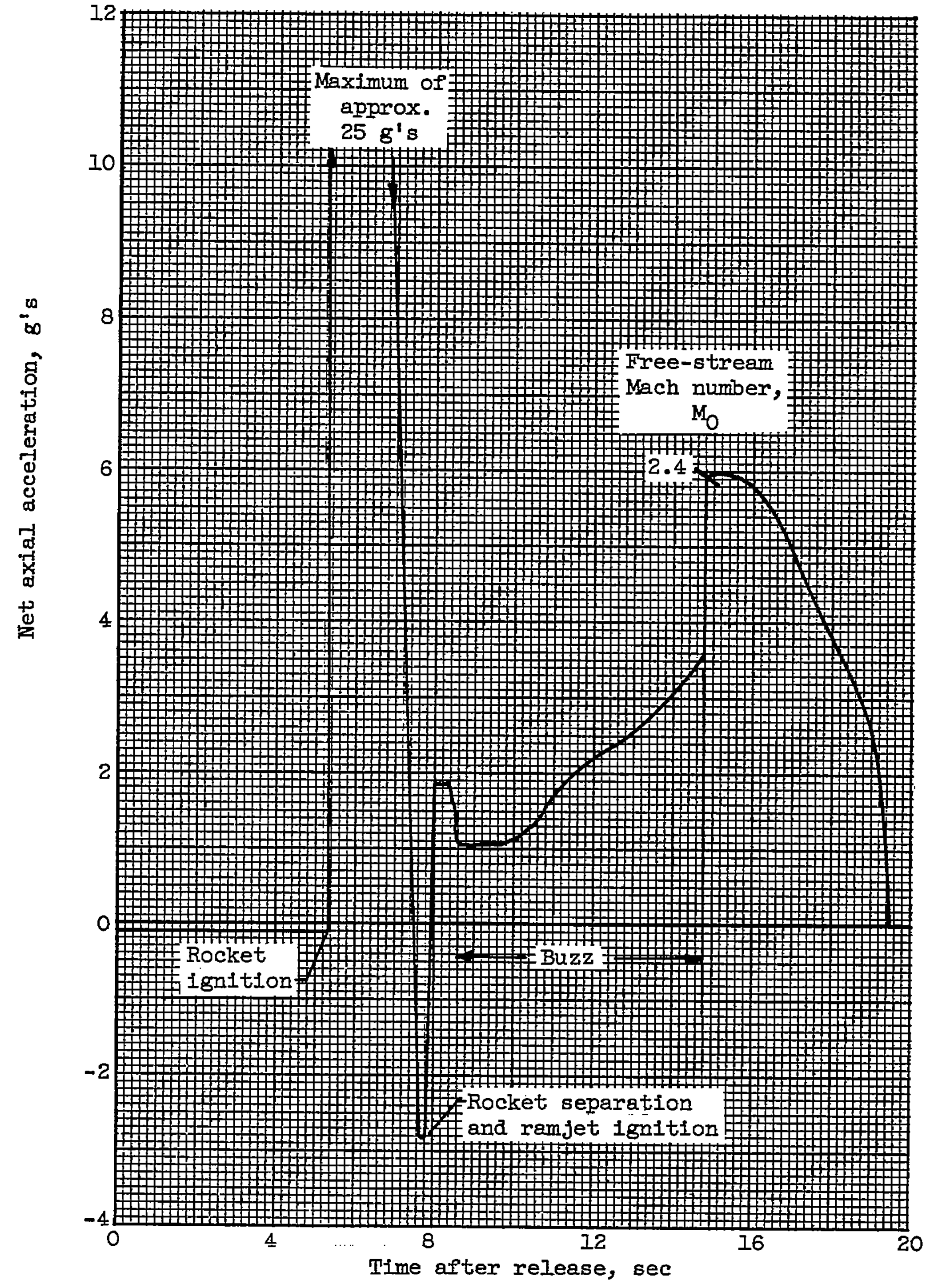
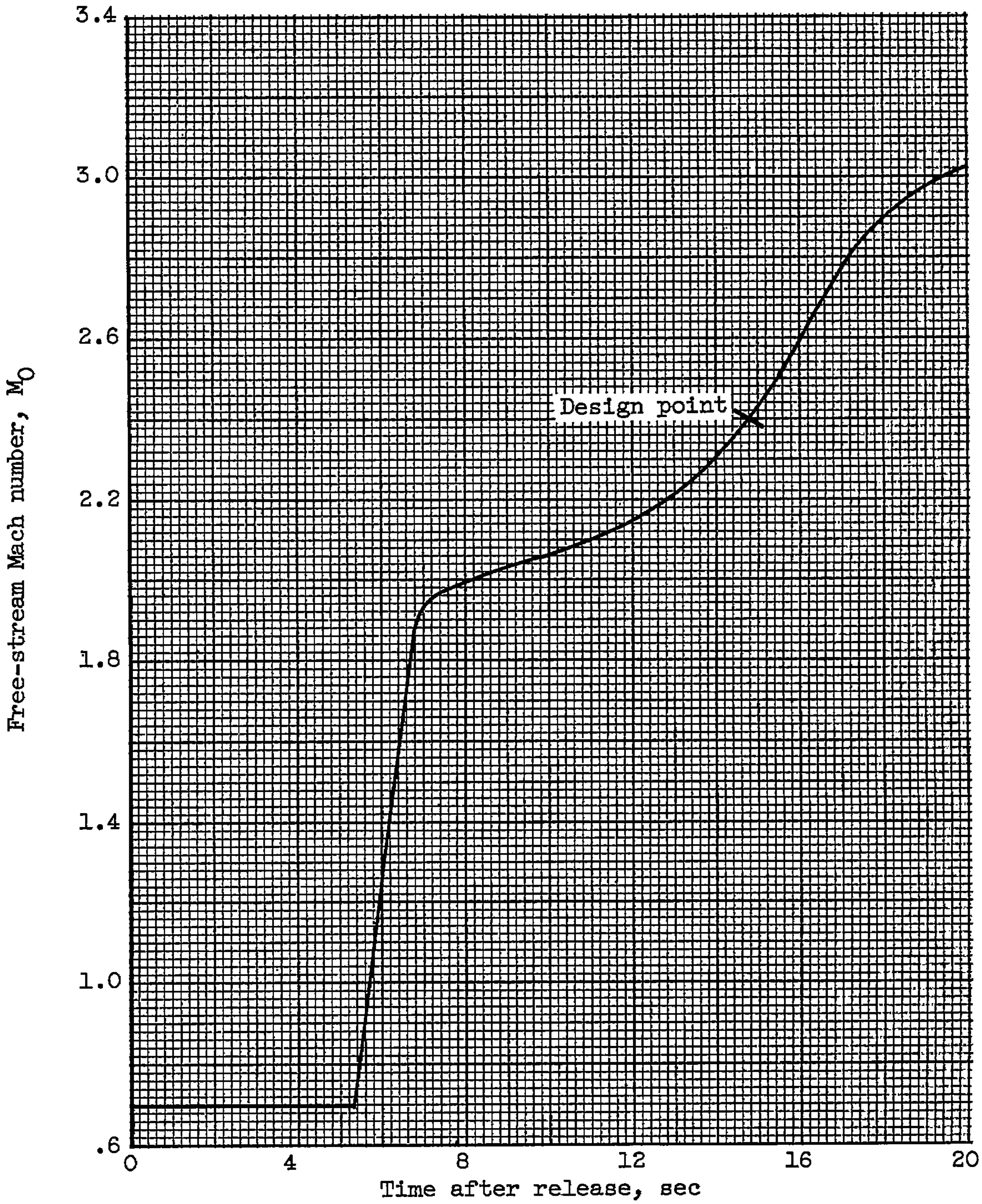
Figure 5. - Front and rear views of flameholder.



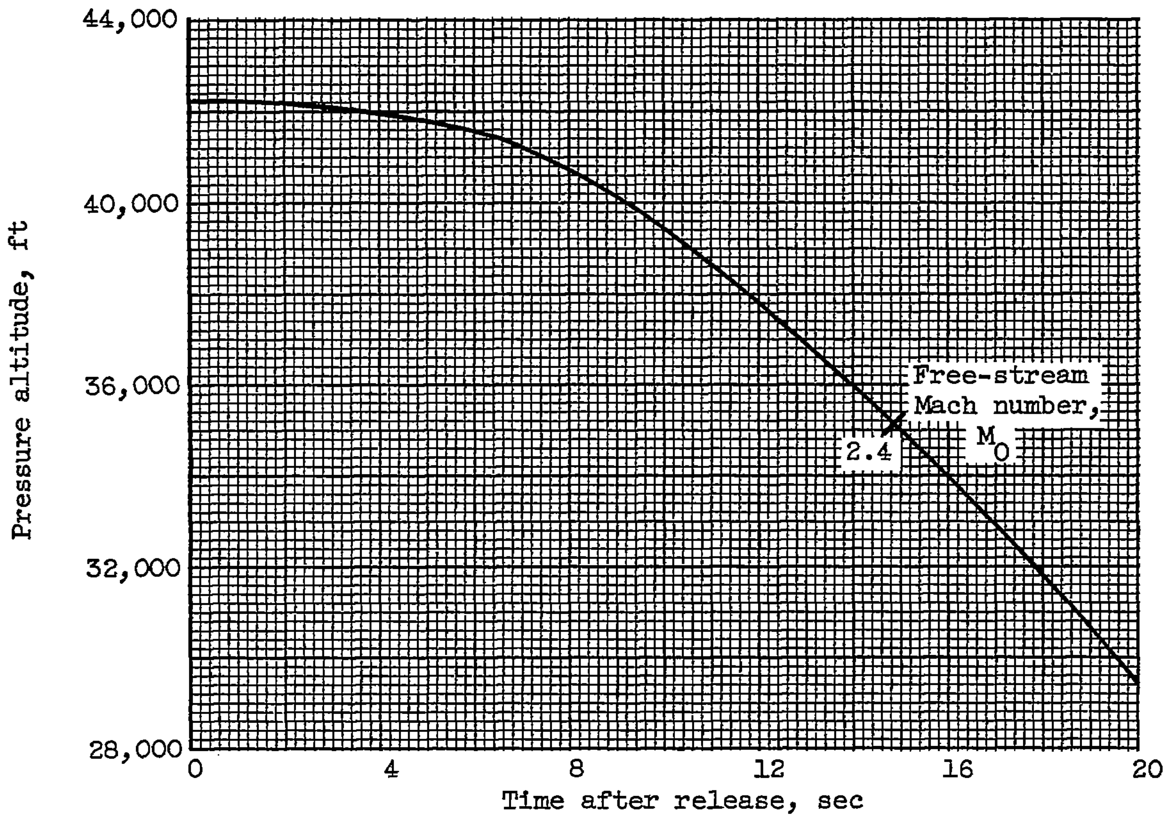
(a) Variation of axial acceleration with time.

Figure 6. - Histories of engine acceleration, Mach number, and altitude.



(b) Variation of free-stream Mach number with time.

Figure 6. - Continued. Histories of engine acceleration, Mach number, and altitude.



(c) Variation of pressure altitude with time.

Figure 6. - Concluded. Histories of engine acceleration, Mach number, and altitude.

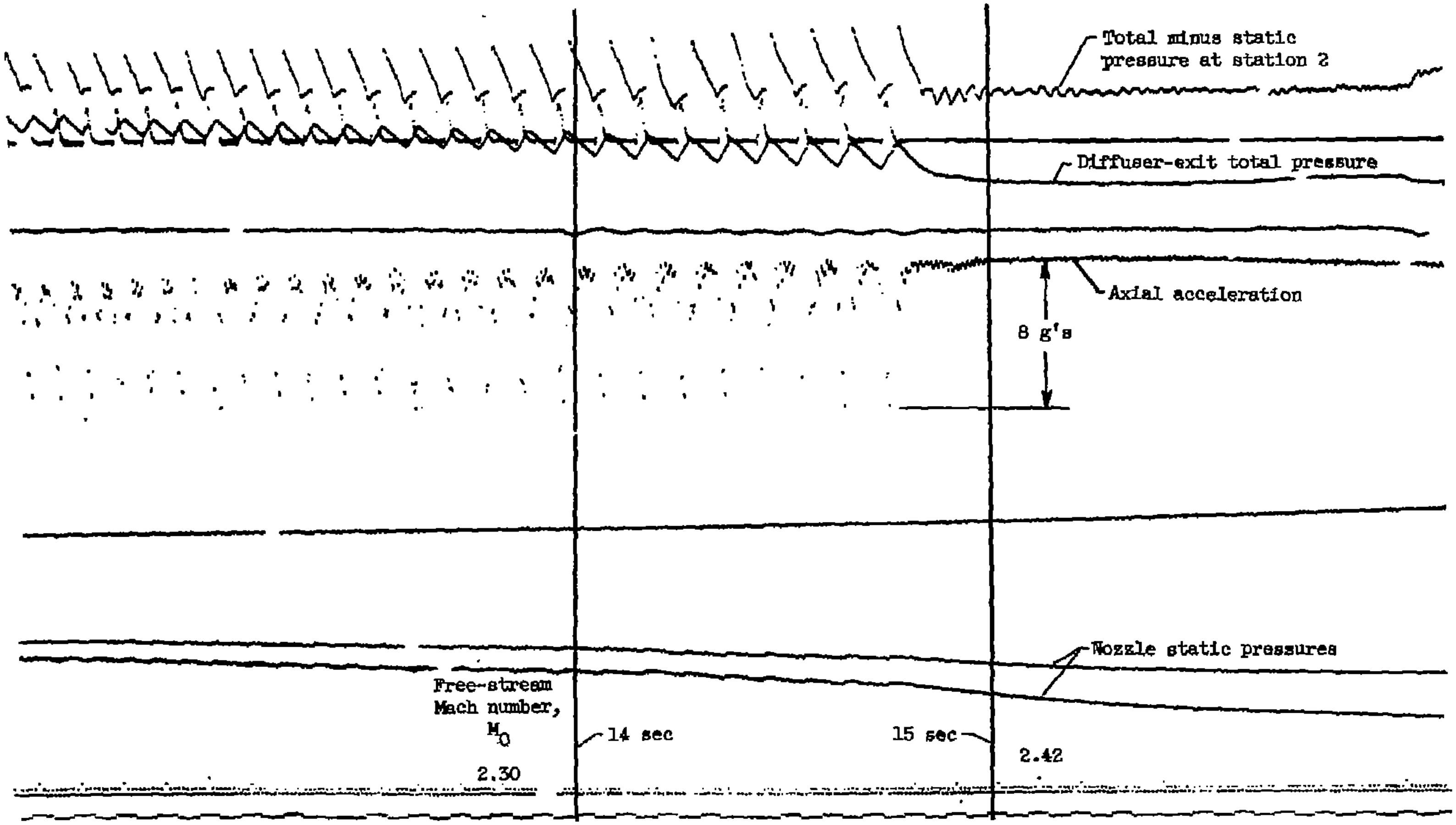


Figure 7. - Variation of acceleration and pressures during and after cessation of buzz.

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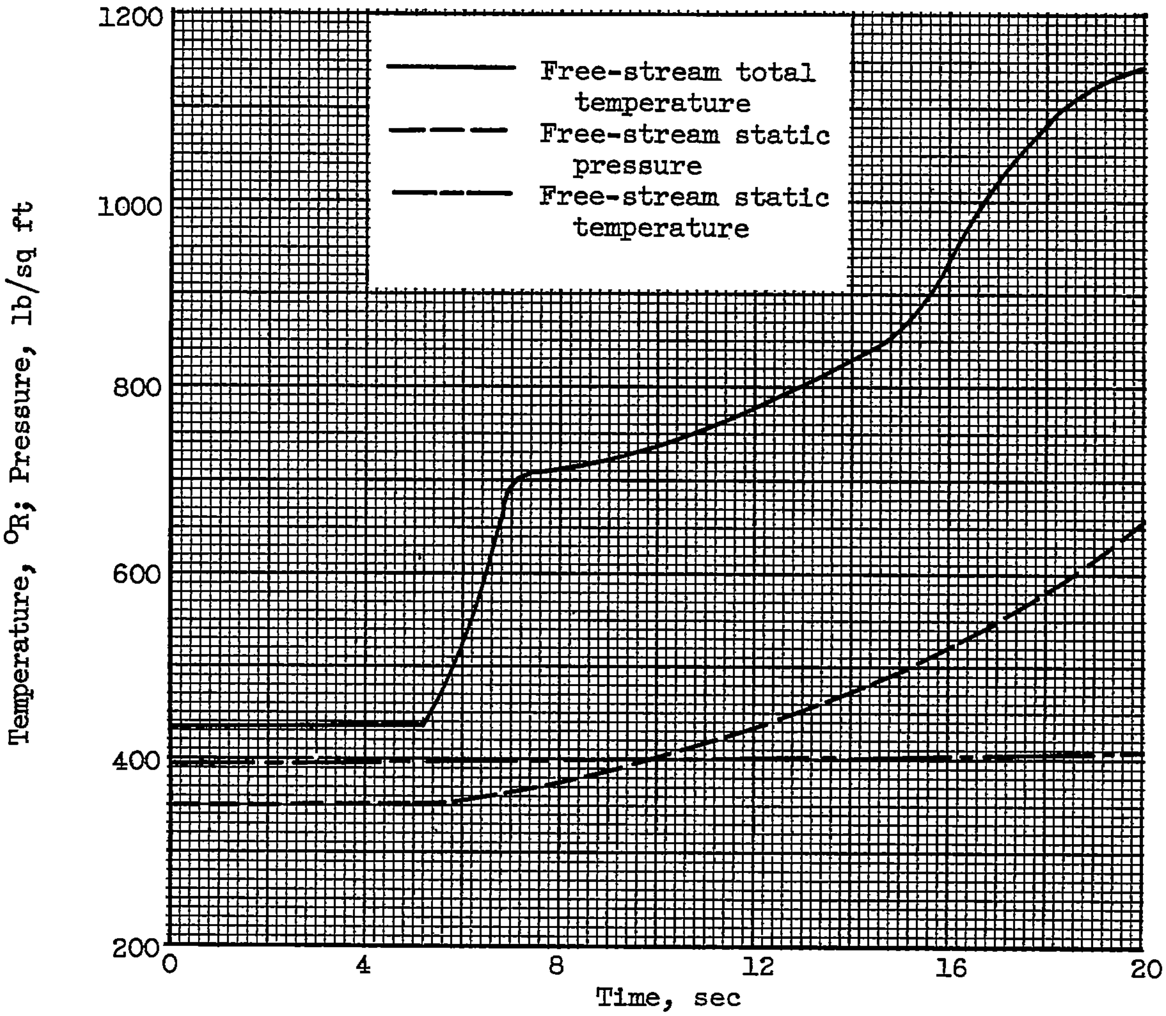

Figure 8. - Variation of free-stream pressure and temperature with time.

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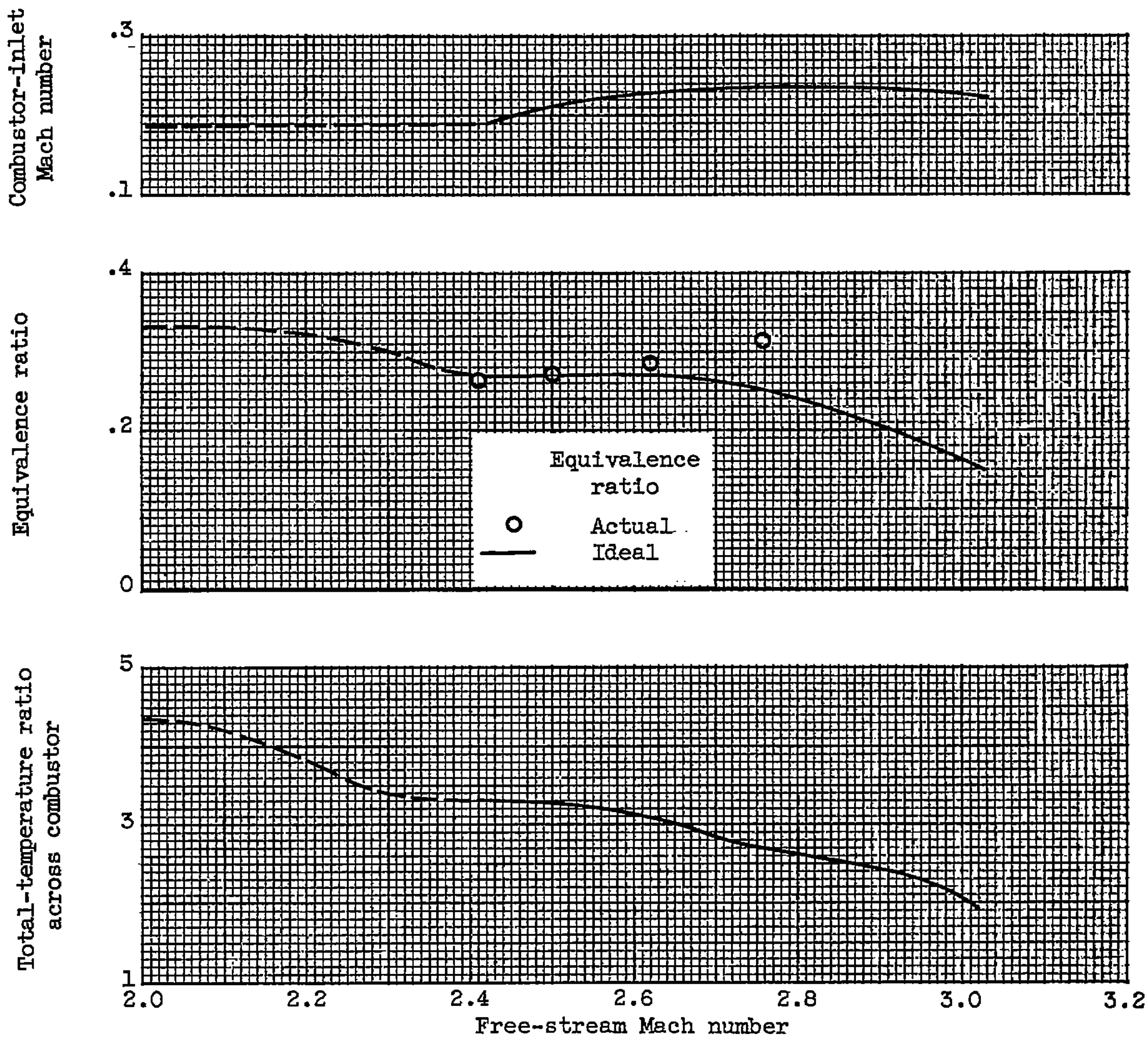

Figure 9. - Variation of combustor conditions with free-stream Mach number. Dashed line denotes approximate values during engine buzz.

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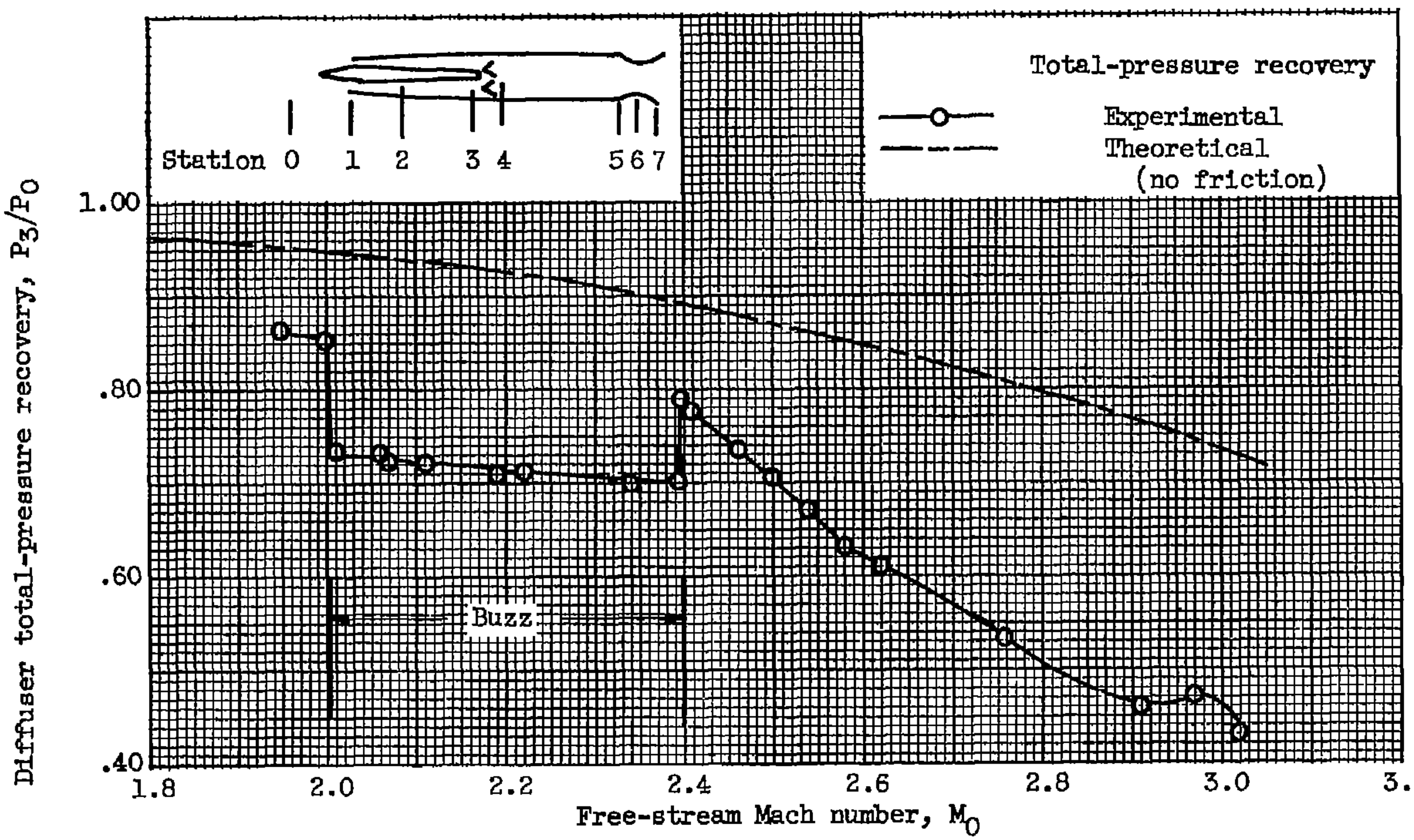

Figure 10. - Variation of diffuser pressure recovery with free-stream Mach number.

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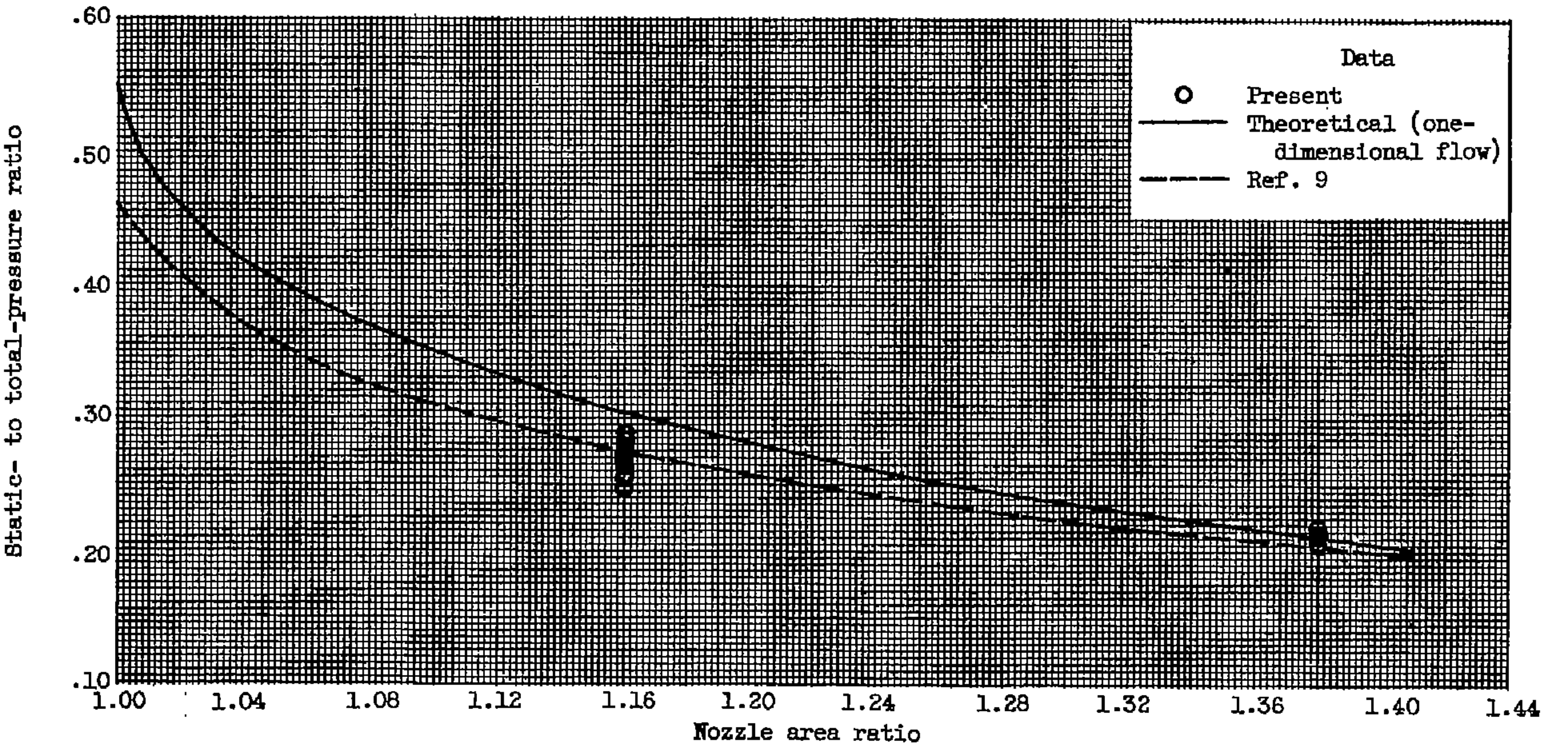

Figure 11. - Comparison of exhaust-nozzle wall static-pressure measurements with theory.

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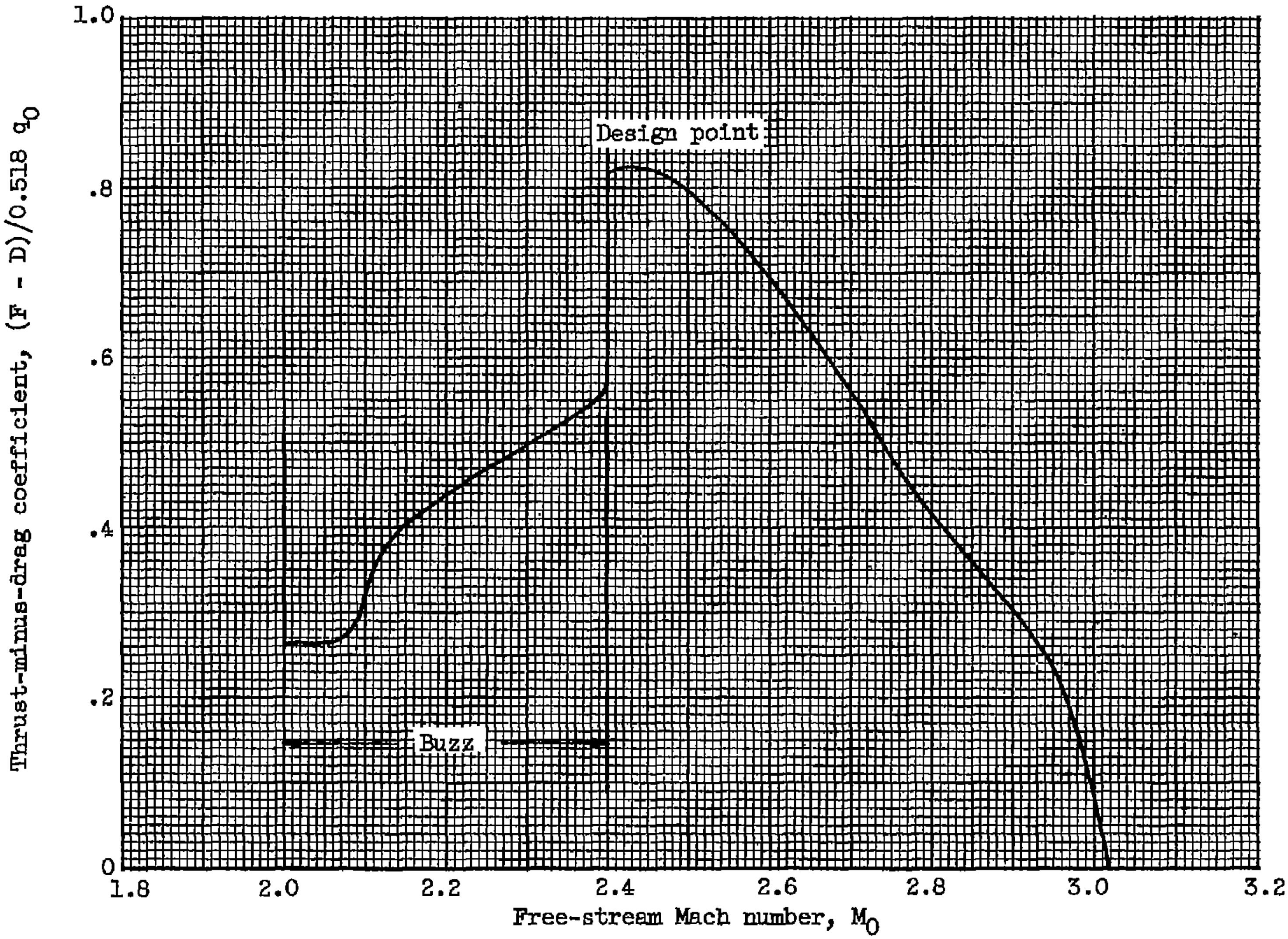

Figure 12. - Variation of thrust-minus-drag coefficient with Mach number.

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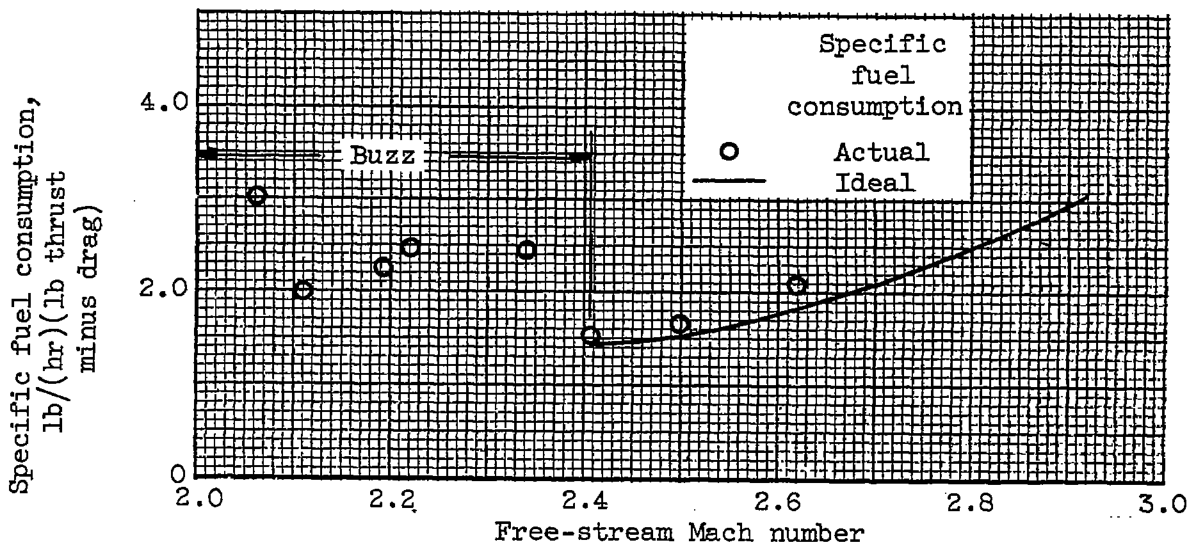

Figure 13. - Variation of measured and ideal specific fuel consumption with free-stream Mach number.

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